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DEVELOPMENT OF A FAIL SAFE DESIGN OXIDATION RESISTANT REINFORCED CARBON SYSTEM FOR THE WING LEADING EDGE OF A SPACE SHUTTLE VEHICLE

NASA-JSC CONTRACT NAS9-12763

PHASE III FINAL REPORT

VOLUME I EXECUTIVE SUMMARY

VSD REPORT NO. T143-5R-30008

VOUGHT SYSTEMS DIVISION
LTV AEROSPACE CORPORATION
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DRD NO. 3

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ABSTRACT

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TITLE OF REPORT					
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INDEXING

A. SUMMARY SENTENCE(S): A preliminary design for a fail safe wing leading edge system for the Space Shuttle is shown together with supporting analyses, component tests and fabrication demonstration.

B. KEY WORDS: Reinforced Pyrolyzed Plastics, Reinforced Carbon-Carbon, Space Shuttle Leading Edge Design, Thermal Protection System, Non-Destructive Testing, Oxidation Inhibited Carbon

WEAPON SYSTEMS NUMBERS, MODEL NUMBERS, ETC.

N.A.

ABSTRACT

A preliminary design of a fail safe wing leading edge system was developed for the Space Shuttle Orbiter. The system design consists of an oxidation resistant reinforced pyrolyzed plastic (RPP) leading edge structure, a support structure and support bracketry to tie the RPP to the aluminum front spar of the Shuttle, and insulation components necessary for the protection of the aluminum spar. The RPP is fail safe in that it is designed to permit one safe entry to landing without benefit of the oxidation protective coating.

Technology was developed for fabrication of 0.5 in. (1.27 cm) thick RPP to support the fail safe concept. Its practicality was demonstrated by the successful fabrication of three full scale leading edges, two of which are part of a two-segment leading edge assembly, and one for destructive test.

Feasibility demonstration tests were conducted on the support bracket design to prove that hot RPP structure could be tied to cool aluminum; on an insulation component to show that light weight rigid insulation can withstand Shuttle dynamic load environments; on seal strip gaps to verify that temperatures in local areas do not become excessive; on a full size RPP component to determine resistance to design acoustic noise levels.

An NDE technique, using portable eddy current equipment, was developed for determining coating thickness on RPP parts both in the shop and on the flight line.

Strength of thick laminate material was determined and compared with strength of two full scale components (one fabricated during an earlier program phase), which were sectioned and tested.

Analyses and tests demonstrated that RPP leading edges are viable for the Space Shuttle application.

FOREWORD

This final report was prepared by the LTV Aerospace Corporation, Vought Systems Division for NASA/JSC Contract NAS9-12763, Development Of A Fail Safe Design Oxidation Resistant Reinforced Carbon System For The Wing Leading Edge Of A Space Shuttle Vehicle. This work was performed under the direction of the Thermal Technology Branch of the Structures and Mechanics Division with Mr. F. S. Coe, III as the Program Director.

The following individuals were directly responsible for performing the program tasks and in the preparation of this final report: Don While - Project Engineer; Bill Agan and Dwain Bennett - Structures; Jim Medford and Wes Whitten - Thermal; Ed Matza, Frank Tarsia and Al Hill - Design; Bob Bost - Dynamics; Dick Rogers and Kelly Adams - Testing; Ike Harder, Don Rogers, Benny Tillison, Dave Shuford, and Bob Scott - Materials and Processes; Lou Karkos - Manufacturing Project Leader; Jack Vought, Zeke Williams, Stan Whitcher and Tom Mays - Manufacturing; Bob Miller - Quality Control Project Leader; and John Fenton and Roy Littlejohn - Quality Control.

This report is prepared in five volumes. Volume I contains a summary of the wing leading edge program. Volume II provides a detailed technical discussion of the Phase III program. It is arranged in accordance with the major tasks performed under this contract during the period 24 April 1972 through 15 June 1973. Volumes III, IV, and V are appendices containing backup material and more detail technical data on certain tasks.

TABLE OF CONTENTS

1.0 INTRODUCTION AND BACKGROUND

1.1 INTRODUCTION

This final report describes the work performed on NASA Contract NAS9-12763, "Development Of A Fail Safe Design Oxidation Resistant Reinforced Carbon System For The Wing Leading Edge Of A Space Vehicle." The orbital stage of the manned space shuttle will have leading surfaces which experience high heating rates and surface temperatures during the boost and entry phases of the mission. To increase confidence in this thermal protection concept, the feasibility of a "Fail Safe" design of an oxidation resistant reinforced carbon leading edge was investigated. This program was initiated on 24 April 1972. The basic program consisted of six tasks. The objectives of these tasks were to:

- o Task 1 - design and develop a "fail safe" oxidation resistant reinforced carbon thermal protection system for the wing leading edge of the shuttle orbiter.
- o Task 2 - select, design, and test insulation required to insure that an interface temperature of 350°F (177°C) is maintained between the leading edge thermal protection design and the wing box.
- o Task 3 - verify by test, thermal model data for the leading edge support design.
- o Task 4 - establish, through analyses and tests, guidelines for the design of the leading edge seal strip.
- o Task 5 - provide thick ply flat laminates from which element test specimens can be prepared for NASA testing.
- o Task 6 - present program findings in executive summary and final reports.

By Amendment/Modification No. 2S to Contract No. NAS9-12763, the following additional tasks were incorporated into the program.

- o Task 7 - physical and mechanical evaluations of specimens removed from Phase II L/E segment
- o Task 8 - development of NDE method to determine coating thickness.
- o Task 9 - examination of variables that might contribute to observed coating crazing and the determination of materials or process changes that might eliminate the crazing phenomena
- o Task 10 - provide additional thick ply flat laminates from which element test specimens can be prepared for NASA testing.
- o Task 11 - provide 13-ply flat laminates from which element test specimens can be prepared for NASA testing.
- o Task 12 - fabricate an additional "Fail Safe" leading edge segment.
- o Task 13 - physical and mechanical evaluations of specimens removed from a Phase III Leading Fdge segment
- o Task 14 - establish physical and mechanical property data for thick laminates.

The report of the work accomplished is prepared in three sections. The first section, Volume I, summarizes significant aspects of the work performed. The second section details each of the above tasks in Volume II. The third section includes the various appendices compiled into three additional volumes to provide more detail treatment of certain of the major subtasks. The appendices consist of reports and in-house documentation used in the conduct of the program.

1.2 BACKGROUND

The RPP leading edge design and materials are the result of Shuttle wing leading edge and belly panel programs conducted by VSD under contract to NASA-JSC and Rockwell International (References 1 through 3). These contracts had direct application to this effort and employed NASA approved Rockwell International design criteria as a basis for development and evaluation of designs and materials.

Phase I of the Wing Leading Edge development program (Reference 1) examined variations of oxidation-inhibited RPP materials and designs potentially suited to the Shuttle requirements, and selected candidates for further development. A follow-on program, Phase II (Reference 2), was concerned with development and preliminary evaluation of a selected material system and design concept. An additional program (Reference 3), conducted under contract to Rockwell International, investigated the feasibility of RPP Shuttle fuselage panels.

Because of the application of the Phase I and II leading edge and belly panel programs to the Phase III program, a brief summary of the objectives, extent of investigations, and findings of these studies are provided to establish the point of departure for Phase III.

1.2.1 PHASE I PROGRAM (Reference 1)

The objective of the Phase I leading edge program was to select candidate RPP material systems and designs for the Shuttle wing leading edge, meriting further evaluation and development. A variety of design concepts were examined including sandwich or solid laminate configurations, multi-layered designs with replaceable components, those involving multiple ribs or intercostals, and those using trussed ribs.

Materials investigations were divided into two classes, substrates and coatings, but ultimately the final test of the materials was their compatibility as a system. Substrates, employing graphite or carbon filaments and high and low modulus fibers in cloth or tapes, were evaluated for strength, fabricability, and coating compatibility.

Exploration of coatings involved carbides and oxides of a number of metals applied by diffusion coating, chemical vapor deposition, flame sprayed overlay, added into the resin during initial makeup of the laminate, or some combination of these. Materials evaluation included tests for strength, oxidation resistance in a furnace, and extensive plasma arc exposure simulating the entry environments.

A number of significant findings relative to RPP design for the Shuttle resulted from these exploratory investigations conducted on the Phase I program. Among these were:

(1) Designs employing solid laminates are superior to sandwich designs because they promote internal cross radiation from the hot stagnation region to cooler areas and thus reduce stagnation temperatures and thermal gradients around the leading edge. This increases coating life and relieves thermoelastic stresses. Sandwich designs suppress cross radiation.

(2) Solid laminate designs are less sensitive to strength degradation from the coating system than their thin-skinned sandwich counterparts. This is because solid laminates are thicker and coating depth tolerance has less deleterious effect on strength variations.

(3) Oxidation inhibitors can be diffused into the substrate to a controlled depth to provide multi-mission capability without weakening the interlaminar strength of the composite. By contrast, oxidation inhibitors introduced into the resin during molding and subsequently reacted to form carbides, seriously weaken the interlaminar properties, and are to be avoided.

(4) Overspray coatings do not adhere well and are limited to one or two entries.

(5) Coating systems based on silicon carbide were found to offer the highest temperature reuse capability of all systems examined.

(6) Silicon carbide coating systems offer low catalycity, a phenomenon that reduces surface temperature of the material in a dissociated flow field entry environment. This effect, which amounts to several hundred degrees

Fahrenheit, can be used to extend mission life or system temperature limits, or alternately, provide higher margin on system life.

(7) Coating compatibility and performance in the coated condition are the measurements of most value in judging acceptability of substrates. High modulus substrate materials are not required and show coating expansion mismatch. Low modulus, low strength, low cost materials of the WCA type employed are acceptable for the leading edge application and provide compatible and reliable substrates.

1.2.2 PHASE II PROGRAM (Reference 2)

Phase II of the leading edge program was devoted to the further development of the silicon carbide coated RPP system and refinement of the design to reflect requirements peculiar to the delta-winged Orbiter. Three coated full-scale leading edge segments were fabricated and were load and thermal tested to verify design/fabrication capability. These segments closely approximate the wing geometry at 55% exposed span of the current Orbiter configuration. Another coated component, representative of wing tip leading edge geometry with small (1.6 inch)(4.1 cm) radius was also fabricated and tested successfully in the NASA-JSC 10 MW plasma arc facility. Significant accomplishments from the Phase II program included:

(1) Demonstration of the producibility of large (26 in. x 15 in.) (66 cm x 38 cm) curved panels with smooth surfaces and tolerance control within the ± 0.03 in. (0.08 cm) allowance, and small (9 in. x 7 in. x 5 in.) (23 cm X 18 cm x 13 cm) leading edge segments. These are illustrated in Figures 1-1 and 1-2.

(2) Development of materials property data that showed acceptable room temperature strength which increases with temperature, and a surprisingly high resistance to fatigue loading.

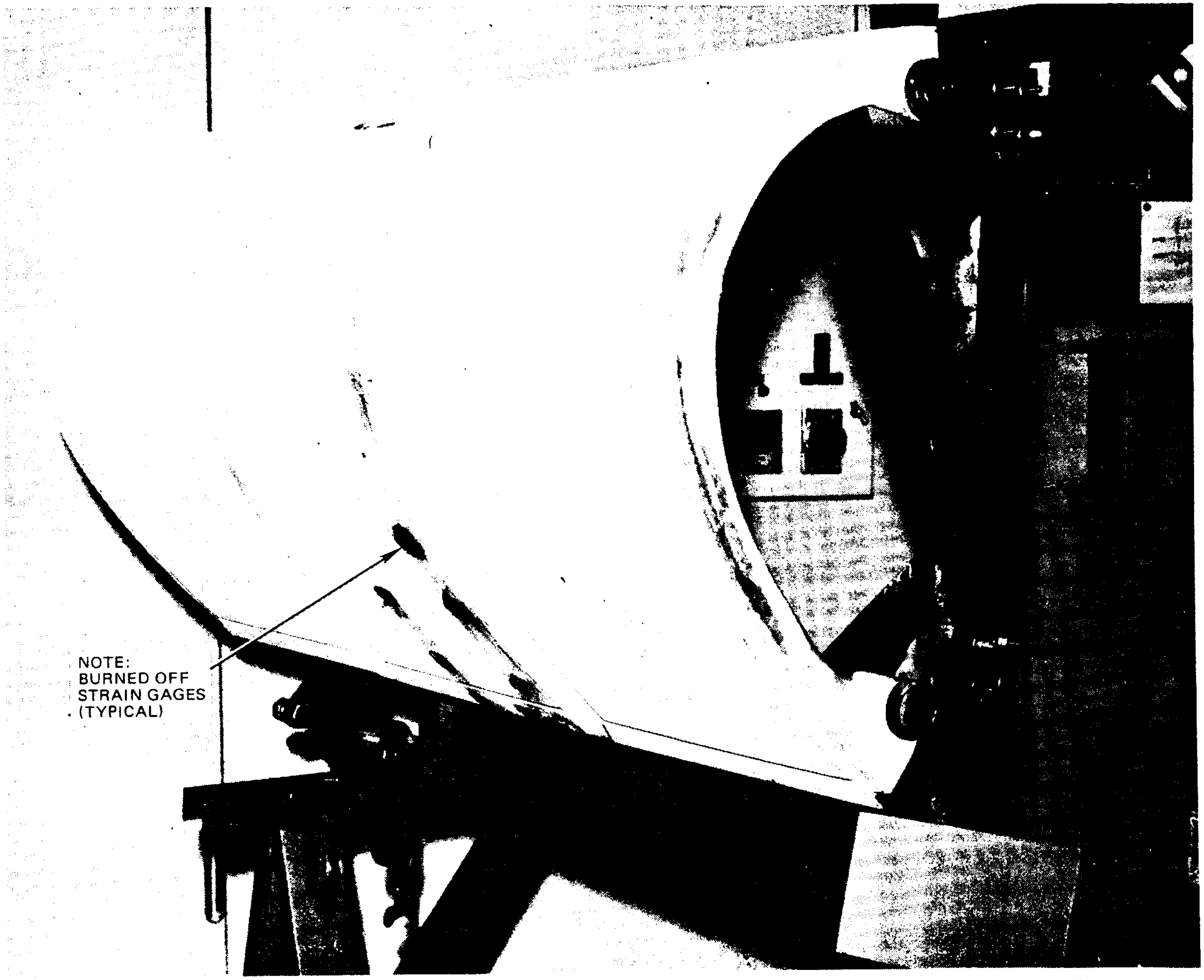


FIGURE 1–1 PHASE II WING LEADING EDGE – FULL SIZE THREE-SEGMENT ASSEMBLY

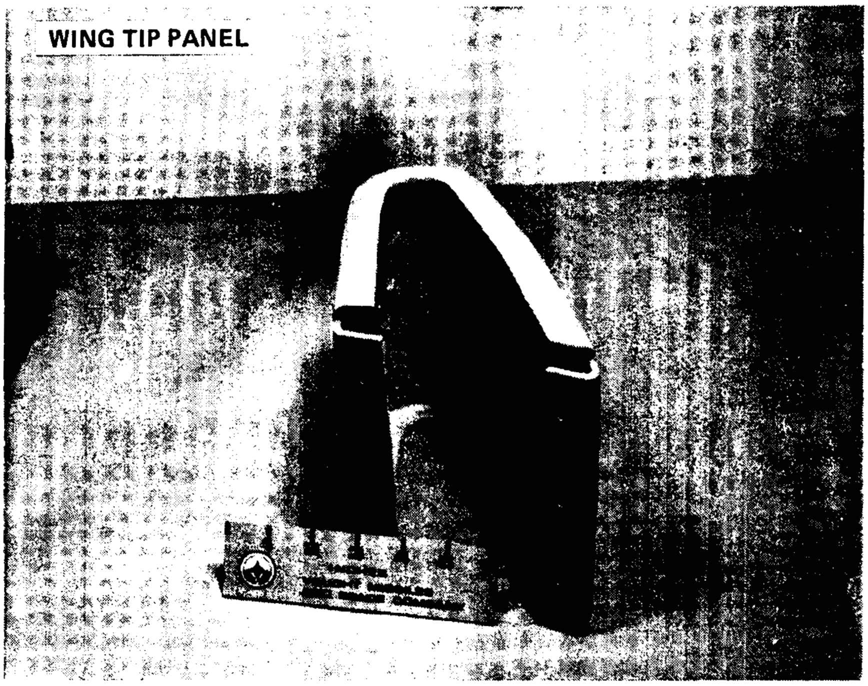


FIGURE 1–2 PHASE II LEADING EDGE SUBSCALE TEST MODEL

(3) Demonstration that bonding and riveting (RPP rivets) of RPP components is a practical approach to the design and fabrication of complex structures, when properly employed to prevent bond tension and peel stresses.

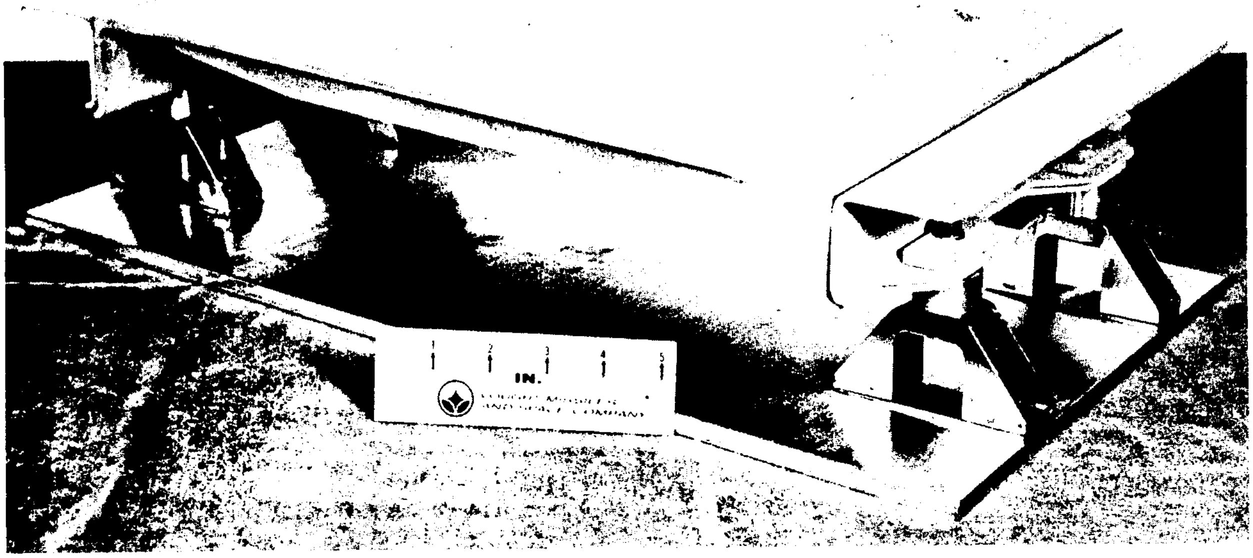
(4) Demonstration that X-ray radiography and ultrasonic-through-transmission non-destructive evaluation techniques are acceptable methods for identifying defects during component fabrication.

(5) Improvement in the science of coating to develop a deeper understanding and gain better control of the coating process. Elements of this technology include raw materials procurement, processing details, and retort design.

1.2.3 BELLY PANEL PROGRAM (Reference 3)

The belly panel program conducted by VSD for Rockwell also added to the experience and data bank on RPP material and designs. The objective of this program was to determine the feasibility of oxidation inhibited RPP Shuttle belly (fuselage) panels, and to develop designs meeting the Shuttle requirements. The test demonstration was successfully completed by Rockwell International on a delivered panel assembly (Figure 1-3) and verified that RPP is indeed a feasible fuselage panel material. The RPP proved weight competitive with metallic TPS, which was still under consideration at the time, to temperatures from 2500°F (1371°C) to below 1800°F (982°C).

The belly panel program examined a number of panel configurations and associated support structure, selected several concepts for component joint tests, and then selected a final configuration for fabrication and test. Panel configurations examined included integrally fabricated "I", "T" or hat-stiffened panels and double-skinned or boxes panels. Joint concepts, designed for panel installation from outside the vehicle, included slip joints, flexure joints, those requiring refractory fasteners and those wherein the fasteners were buried to permit the use of super alloys.



DESIGN

- TEMPERATURE – 2500°F (1371°C)
- BURST PRESSURE – 4.5 PSI ULT (3.1×10^4 N/M^2)
- COLLAPSE PRESSURE – 3.6 PSI ULT (2.5×10^4 N/M^2)

TEST

- LIMIT BURST AND COLLAPSE PRESSURE
- THERMAL CYCLE – 100 HIGH CROSS RANGE SIMULATED MISSIONS (INERT ATMOSPHERE)
- ULTIMATE (4.5 PSI) BURST PRESSURE (3.1×10^4 N/M^2)
- NO FAILURE

FIGURE 1–3 FUSELAGE TEST PANEL

Fabrication of slip and flexure joints, and "T", "I" and hat section panels for joint tests showed that integrally fabricated supports using the flexure principle (flexible supports, as shown in Figure 1-3) for thermal expansion stress alleviation was a practical approach. This concept was tested satisfactorily under cyclic load and was the only one of four panel/joining concepts tested that did not develop "slop" in 100 cycles of limit load. This program showed that:

(1) Sophisticated oxidation inhibited RPP hardware can be fabricated within the desired ± 0.03 in. (0.08 cm) tolerance, comparable to fiberglass components.

(2) Panels designed with integral support legs can be fabricated and offer an excellent solution for panel anchoring and relief of thermal expansion, where external access for installation is required.

(3) Oxidation-inhibited RPP shows a resistance to catastrophic structural failure from local damage. This was demonstrated in limit load cycle joint tests, where in two examples, it was noted that total failure did not occur at the first sign of local damage.

1.2.4 SIGNIFICANCE OF THESE CONTRACTS

These contracts showed that:

(1) Complex RPP hardware in the sizes required for the Space Shuttle Orbiter are practical.

(2) Solid laminate designs are the best solution.

(3) Silicon coating systems are the best available and provide multi-mission capability.

(4) Existing, relatively low-cost graphite cloth reinforcement material is compatible with the coating system and is suitable for Shuttle RPP components.

2.0 SUMMARY

The Phase III program was built upon the earlier two phases and had three fundamental tasks to be addressed:

(1) Development of a totally integrated leading edge design from the wing front beam forward. This design would include design details and materials selection for the support structure and insulation necessary for a workable RPP concept on the Shuttle.

(2) Development of a process for thick (0.5 in.) (1.27 cm) laminate fabrication to permit one entry of the baseline wing leading edge without benefit of the oxidation protection coating.

(3) Development of a non-destructive evaluation (NDE) technique that permits thickness measurement of the oxidation protection coating between flights.

Each of these tasks was accomplished successfully, fortifying the concention that coated RPP is viable for the Shuttle application. A number of other tasks were accomplished during the conduct of the program in support of the major tasks. Among these were materials performance data gathering, reduction of coating crazing, and tests to verify design concepts. Each task is discussed in sequence in Volume II. This summary covers only major highlights of the program, as follows:

- Design description
- Proof of design
- Fabrication
- Coating thickness measurement
- Coating crazing reduction
- Destructive evaluation of leading edge segments

2.1 DESIGN DESCRIPTION

The design of the leading edge is a continuation of the design concept developed in the Phase II program and, in fact, utilized identical geometry, design loads, and environments. One difference is that in Phase III the RPP was designed to be "failsafe" in that thicknesses were established to permit one safe entry to landing without benefit of coating on the exterior. This philosophy was imposed as a contingency in the event it was infeasible to determine coating thickness remaining prior to flight. However, through work performed on this program, coating thickness can now be reliably measured using portable eddy current equipment.

Another change from Phase II philosophy was that wing front beam structure would be limited to $350^{o}F$ ($177^{o}C$) for aluminum, rather than the previous titanium limit of $600^{o}F$ ($316^{o}C$). This affected the design of the support bracketry and insulation system.

The wing leading edge design consists of RPP segments attached to a simulated wing support structure at four attachment locations. This is illustrated in Figures 2-1 and 2-2, where the deliverable hardware is shown in two stages of assembly; and in Figure 2-3 where the isometric drawing depicts the structural assembly. A segmented leading edge with expansion joint gaps is necessary to accommodate differential thermal expansion between the hot RPP and the cool aluminum wing structure. These expansion gaps are covered by seal strips (Figure 2-4) to prevent direct flow of hot boundary layer gases into the leading edge cavity. The RPP segments are relatively simple, consisting of the skin panel, closing ribs, and trailing edge spars, all integrally formed into what is essentially a five-sided, curved, boxlike configuration.

Attachment design employs stand-off support fittings and a lug bolt joint at the RPP connection. The lug bolts are located a sufficient distance inside the outer contour (Figure 2-4) to reduce temperature to $1400^{o}F$ ($760^{o}C$) and permit metallic attachment hardware (Inconel 718). In the plane of the ribs the leading edge attachment scheme requires no slip joints because of the inherent flexibility of the leading edge shape, the low expansion coefficient of RPP, and the strength capability of RPP which result in manageable thermoelastic stresses.

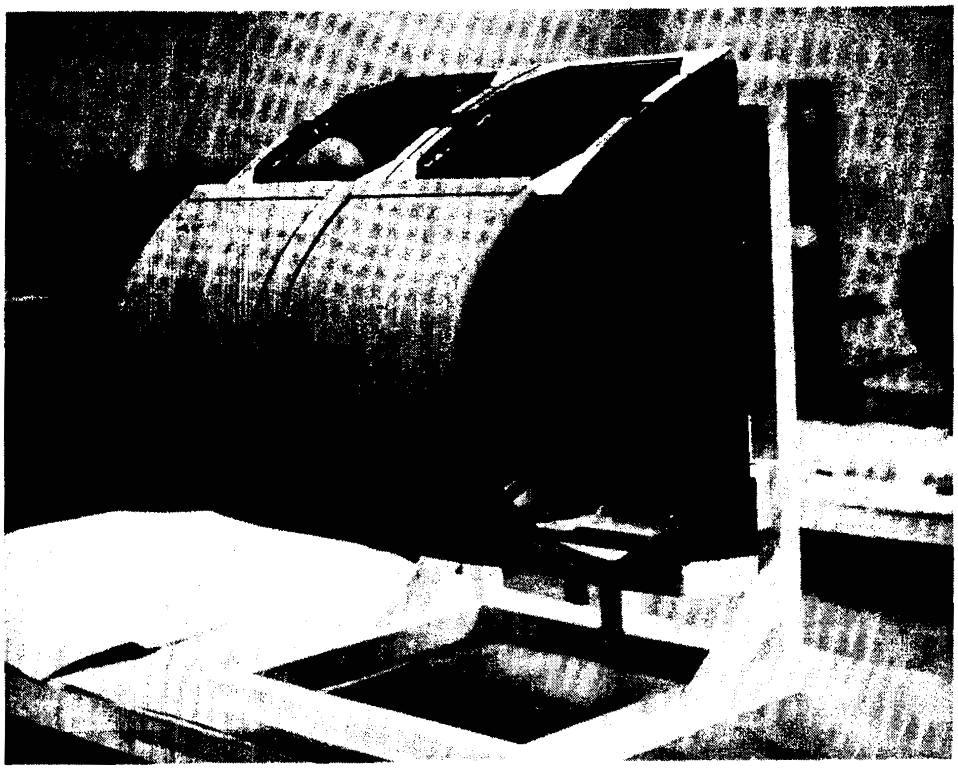


FIGURE 2-1 RCC LEADING EDGE ASSY

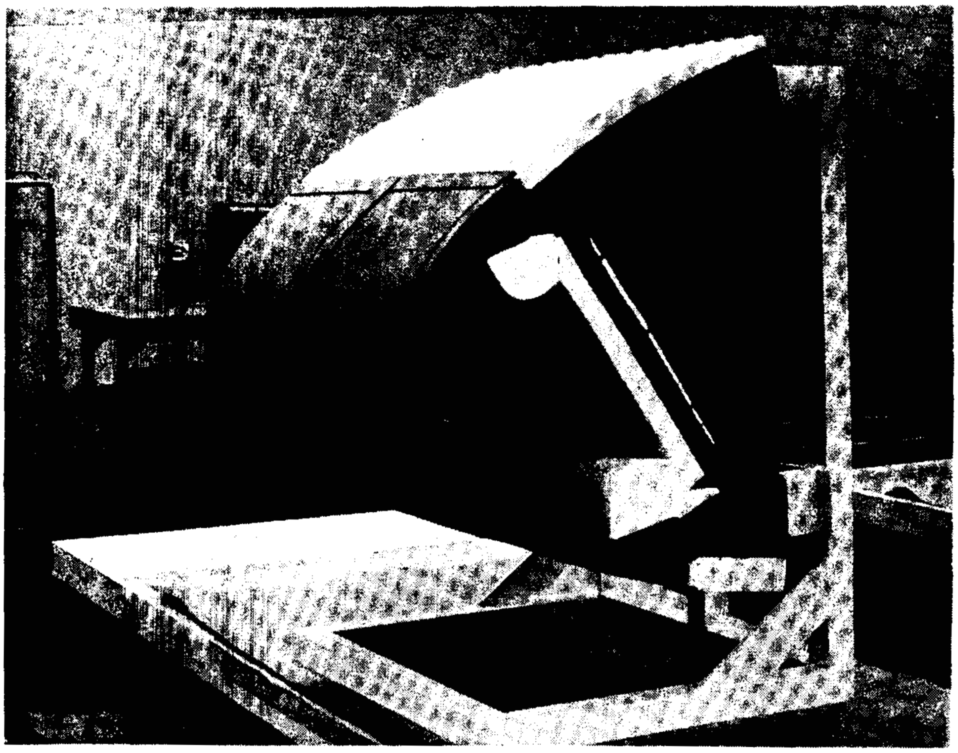


FIGURE 2-2 RCC LEADING EDGE ASSY WITH INSULATION

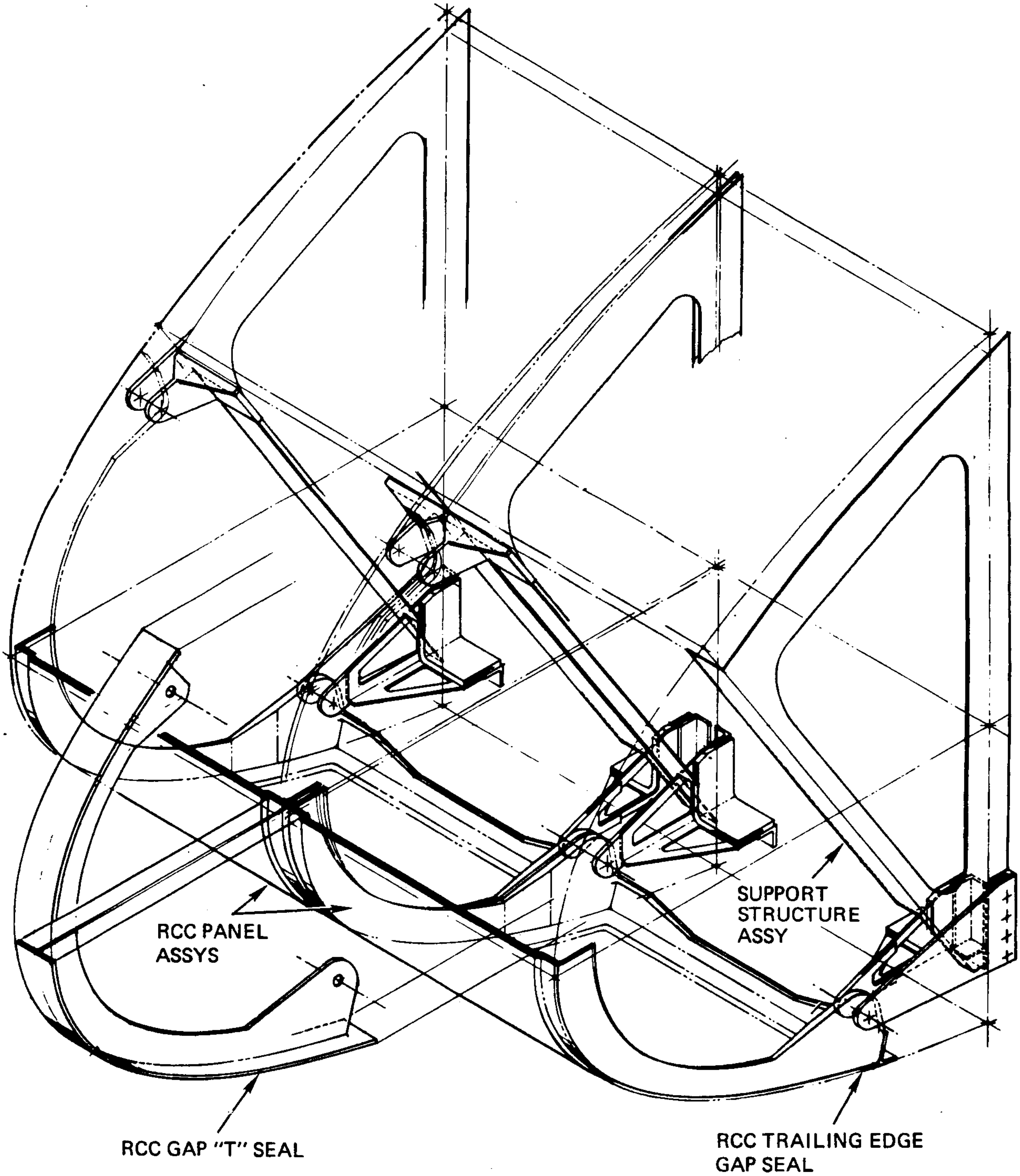
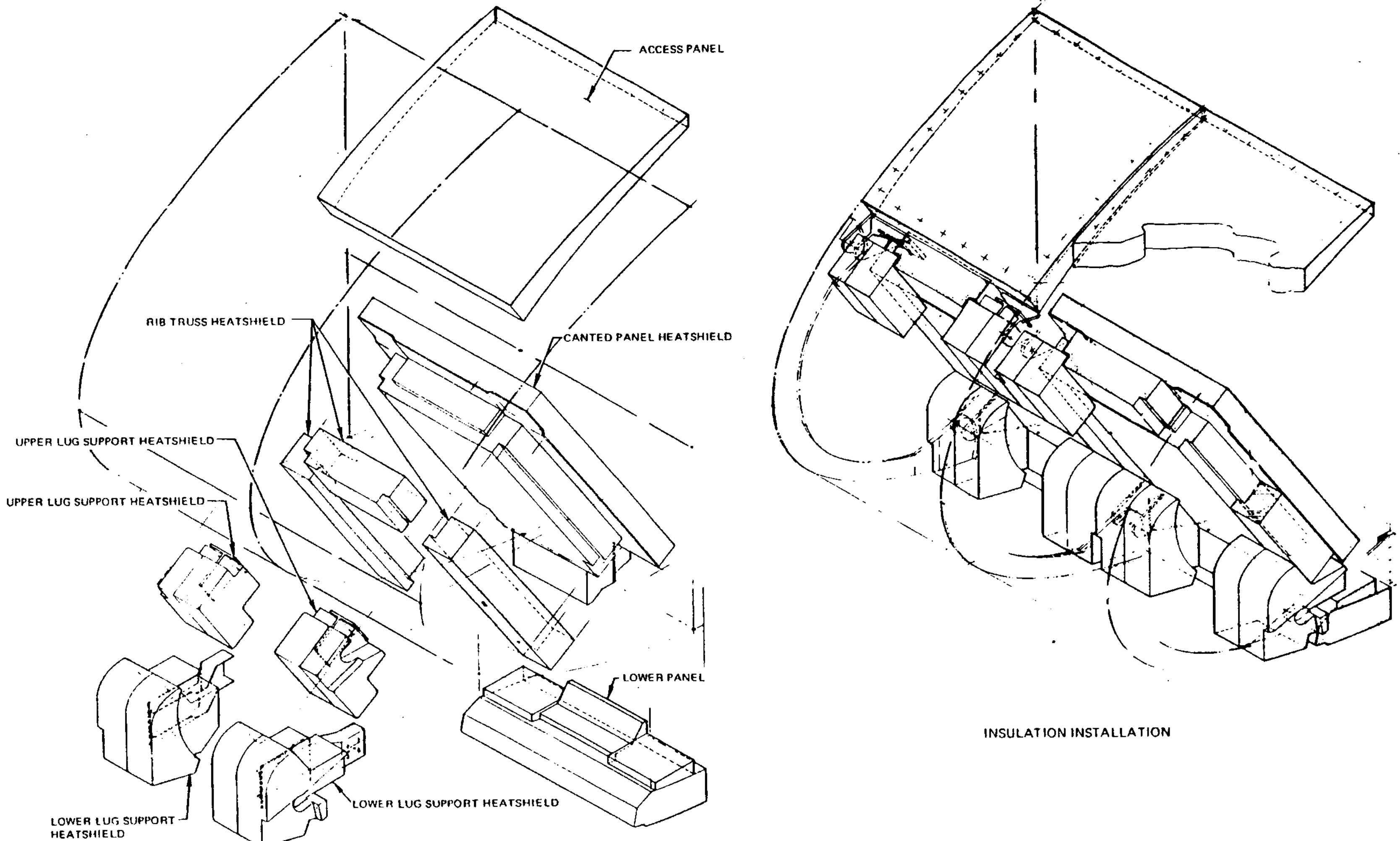


FIGURE 2–3 WING LEADING EDGE STRUCTURAL ASSEMBLY



INSULATION COMPONENTS

INSULATION INSTALLATION

FIGURE 2–3 WING LEADING EDGE STRUCTURAL ASSEMBLY (CONT.)

FOLDOUT FRAME

FOLDOUT FRAME

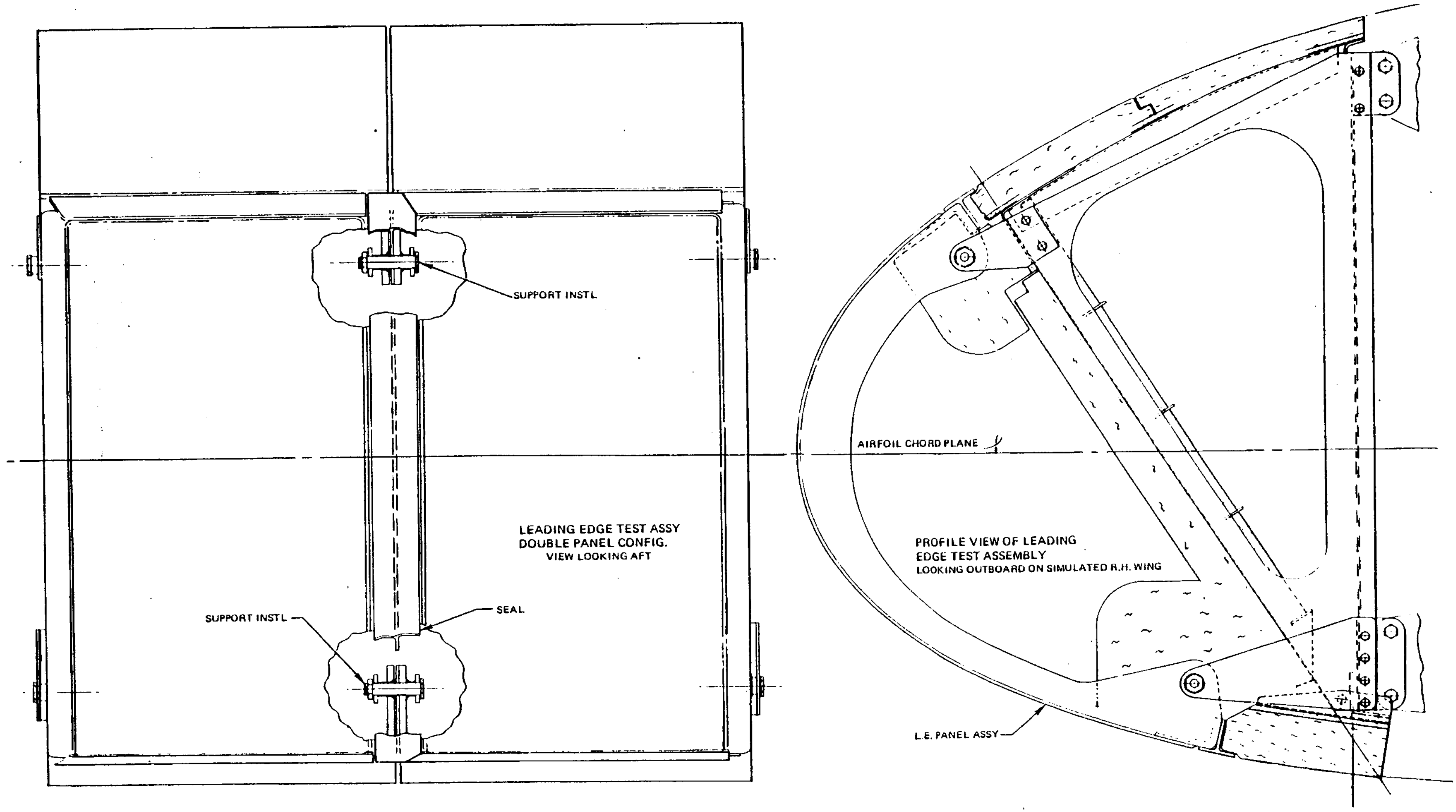


FIGURE 2–4 PHASE III WING LEADING EDGE FULL SIZE TWO-SEGMENT ASSEMBLY

FOLDOUT FRAME

FOLDOUT FRAME

Spanwise expansion is accommodated in the following manner. The outboard side of each segment and its seal strip are held fixed at the lug joint, while the inboard side is permitted to slide on the lug bolt to relieve any tendency for thermal stress buildup from external restraint. This technique worked successfully in the Phase II entry temperature test (Reference 2).

Support fittings are designed to reduce the temperature at the RPP attachment bolt, (which reached 1400°F (760°C) to a maximum aluminum front spar soak temperature of 350°F (177°C). The lower fitting is the most critical for the temperature problem. An Inconel 718 truss type fitting was employed, which provided long heat conduction paths and small conduction area for maximum thermal resistance. This metal fitting transitioned into a polyimide fiberglass insulating fitting at the aft end to affect the final temperature drop to the aluminum. In the Phase III tests, the polyimide fiberglass did not exceed 550°F (288°C), but the capability of this material is in excess of 650°F (343°C) for 100-mission life at the low operational stresses involved.

The upper support fitting is located in a cooler region and ties to a 600°F (316°C) titanium structure. Thus, the design conditions are not nearly as severe as the lower, and neither fiberglass nor a truss fitting are required.

The support structure is a titanium truss and forms the tie between the upper support fittings and the simulated front beam.

The insulation assembly is also shown in the figures. Removable insulation is required around each support fitting to gain access to the lug bolts. A removable canted heatshield is required for access to the leading edge cavity. Also shown are upper and lower panels which simulate external RSI panels. Interior insulation for the Phase III test article is a combination of 10 lb/ft^3 and 15 lb/ft^3 (0.16 and 0.24 gm/cm^3) Dynaquartz silica insulation with an operating temperature limit in excess of 2500°F (1371°C). RSI was the preferred material but cost and unavailability precluded its use for the technology program.

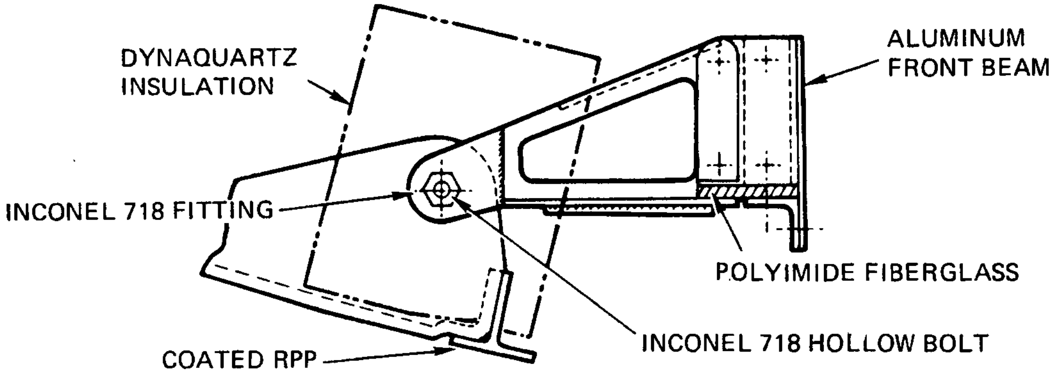
Two single segment assemblies were fabricated for NASA-JSC test. These units are designed to be combined into a single two-segment assembly, Figures 2-1, 2-2 and 2-4 for thermal test.

2.2 PROOF OF DESIGN

Two elements of the design were of particular importance and were examined by analysis and test. These were: (1) lower support lug thermal design to ensure that the hot RPP could be realistically attached to cool aluminum structure, and (2) seal strip gap heating. The support lug test proved that surface temperatures as high as 2100°F (1149°C) can be reduced to 231°F (110°C) maximum at the aluminum spar for the design entry temperature profile. Results of two separate tests of the attachment joint are shown in Figure 2-5. The higher temperatures of Test No. 1 were obtained when input temperatures at the RPP and insulation front face exceeded desired control values.

Gap heating tests were conducted on a bare RPP and graphite model having 1.6 in. (4.06 cm) leading edge radius, in the NASA-JSC 10 MW plasma arc. The "T" seal and gap geometry represented full-scale dimensions. Test conditions were selected to produce calculated flight vehicle boundary layer thickness on the 60° swept model. Typical test results for different configurations with varying gap geometry, representing 12 separate plasma arc tests, are summarized in Figure 2-6, while the test model is shown in Figure 2-7. This data indicated that the "T" seals will indeed become hotter than in the undisturbed regions, but one of the major reasons for this is the blockage of backside radiation. The total excursion of temperature at a given point in the various tests is relatively small despite different gaps and rounding of "T" seal corners. Much of the temperature excursion could be in measurement tolerance. With some design refinements to decrease gaps and conduction paths, the "T" seal temperatures should become more manageable.

An acoustic test, designed to evaluate the structural integrity of the panel and lug holes, was also conducted. The leading edge panel was



LOCATION	MAX TEST TEMP °F (°C)		ALLOWABLE TEMP °F (°C) MAX
	TEST #1	TEST #2	
COATED RPP	2360 (1294)	2090 (1144)	3000 (1649)
INSULATION FRONT FACE	2650 (1454)	2330 (1278)	2500+ (1371)
INCONEL BOLT	1370 (743)	1320 (715)	1700 (927)
INCONEL FITTING	1270 (688)	1220 (660)	1700 (927)
POLYIMIDE FIBERGLASS	550 (288)	450 (232)	650 (343)
ALUMINUM BEAM	255 (124)	231 (110)	350 (177)

FIGURE 2–5 SUPPORT LUG THERMAL TEST SUMMARY

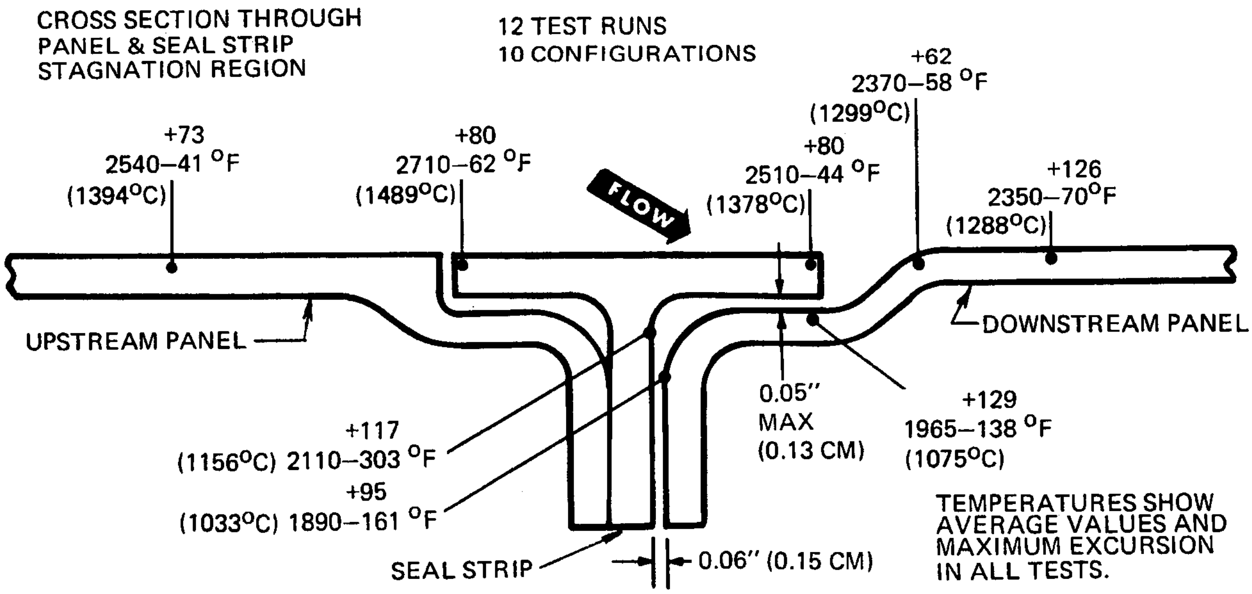
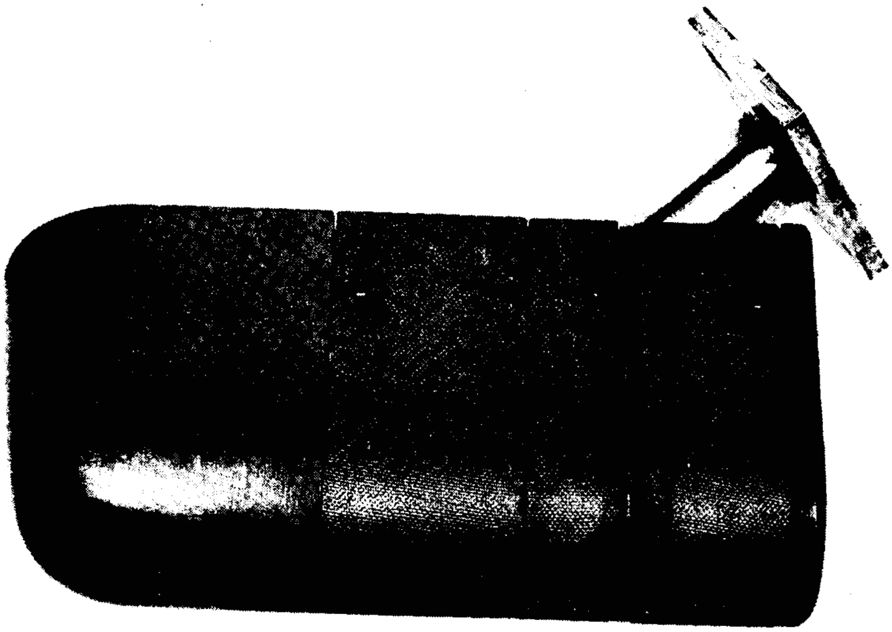
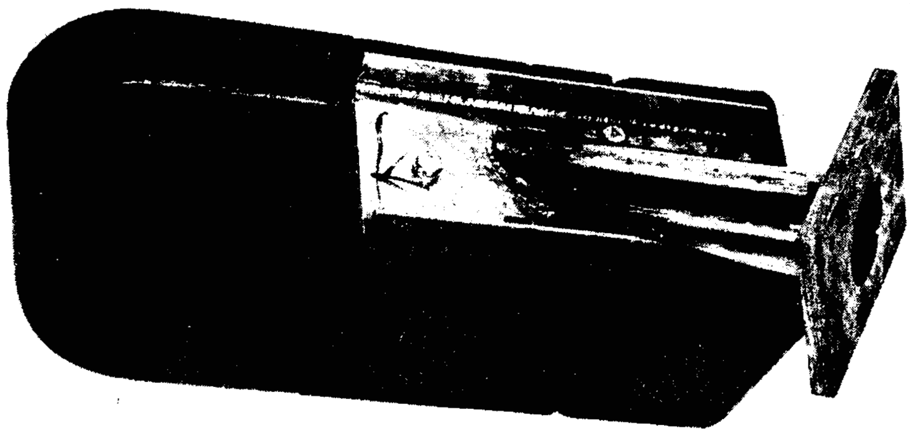


FIGURE 2–6 SEAL STRIP GAP HEATING TEST



FRONT VIEW



REAR VIEW

FIGURE 2–7 GAP HEATING TEST MODEL

mounted in a progressive wave acoustic facility with the test spectrum representing launch acoustic levels at maximum dynamic pressure. Overall sound pressure level of 163 db was applied for 50 minutes, representing 100 missions.

The first test resulted in enlargement of the chamfered holes in the 0.3 in. (0.76 cm) thick lug region on the free side of the leading edge panel, and minor chipping (not exposing bare RPP) around the holes on the fixed side. No other damage to the panel was evident.

A second test was conducted with beefed up lug holes and two design approaches. The fixed side of the panel employed only thicker (0.5 in. (1.27 cm) on top lug and 0.4 in. (1.02 cm) on bottom) material and 0.75 in. (1.90 cm) diameter holes. On the floating side, split bushings were installed to 0.69 in (1.75 cm) diameter holes in the thickened material so that intimate contact with the RPP was assured. None of the holes, bushed or non-bushed, showed evidence of coating failure after the second 50 minute test. With a total of 100 minutes of testing, all other areas of the panel also remained sound. The component is pictured in Figure 2-8. Destructive evaluation of this test article revealed no apparent strength degradation from the acoustic environment.

Vibration testing of a segment of the canted heatshield was accomplished to demonstrate that a rigid, bonded insulation system could sustain dynamic environments imposed by the Shuttle structure. Two tests were conducted. The first test, which produced fatigue failure in minutes, imposed the initial design requirement of 32 g-rms on the component. This was later determined by NASA-JSC to be excessively conservative and a reduced level was ultimately employed in a retest with a modified component. This successful component, pictured in Figure 2-9, sustained 36 minutes at 1.0 g^2/Hz and 18.5 g-rms and 36 minutes at 2.5 g^2/Hz (27 g-rms) without failure. The 36 minute period was calculated to be equivalent to 100 missions at the natural frequency of the test component compared to that of the complete heatshield assembly. This component employed experimental 15 lb/ft^3 (0.24 gm/cm^3) density Dynaquartz insulation and two layers of 0.25 in. (0.64 cm) thick strain isolator. It indicated that an RSI heatshield assembly so configured would also sustain the imposed dynamic environment.

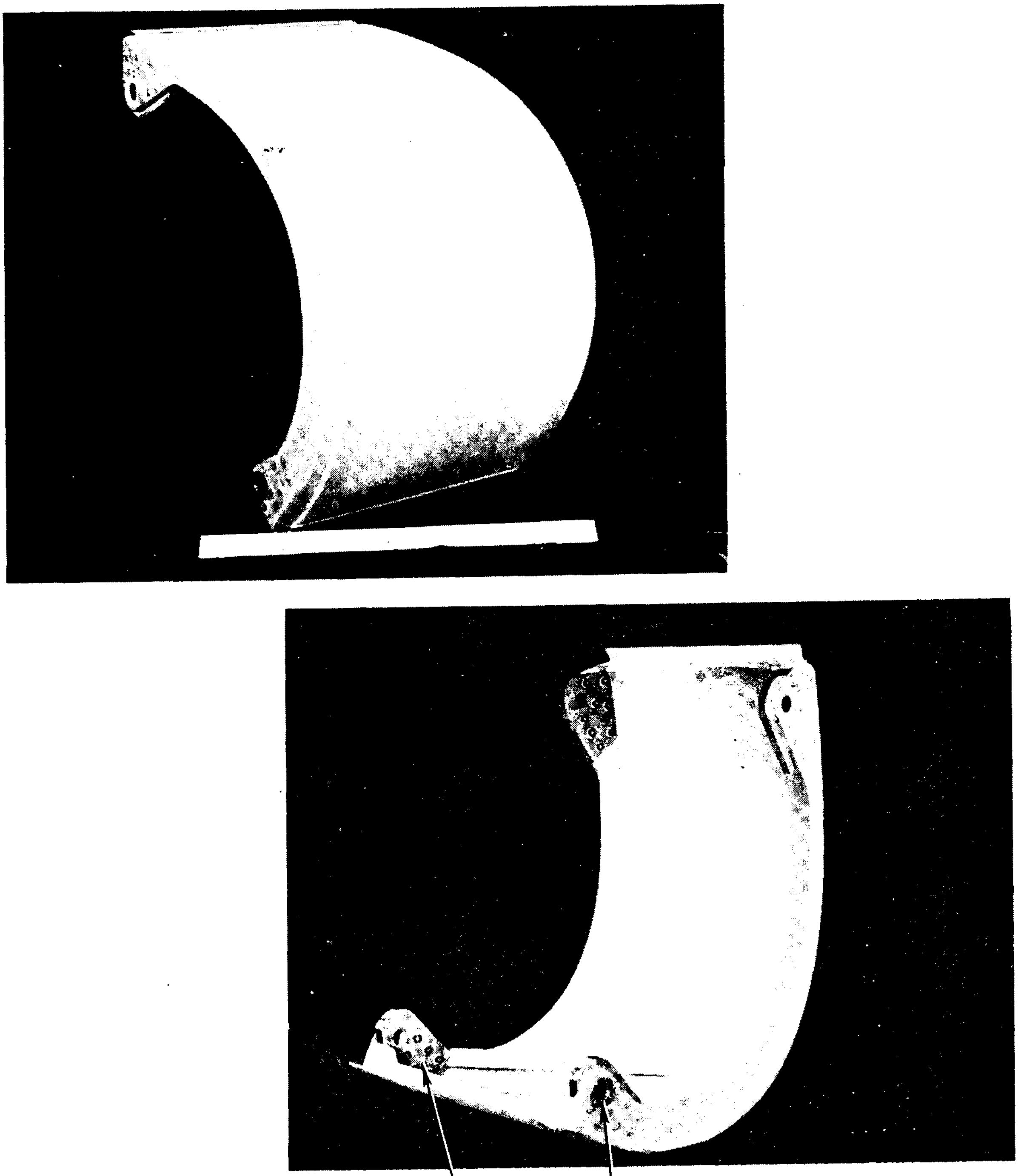
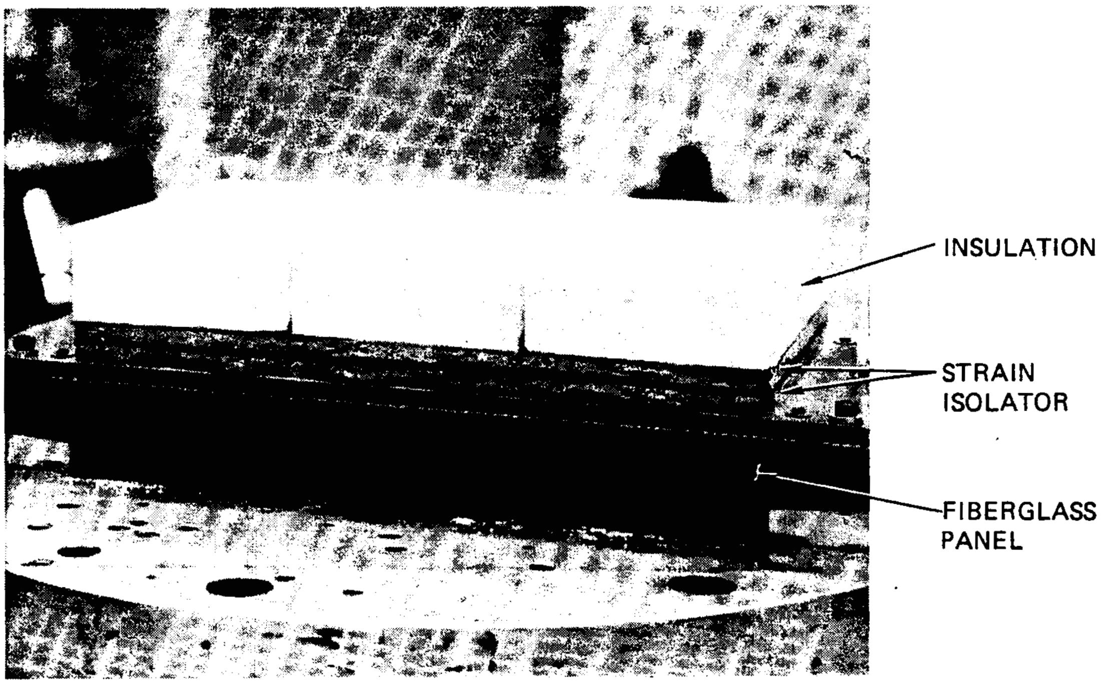


FIGURE 2–8 ACOUSTIC/DESTRUCTIVE TEST LEADING EDGE ASSEMBLY



18.5 G-RMS	27 G-RMS
1 G^2/HZ	2.5 G^2/HZ
36 MIN.	36 MIN.

$\boxed{\text{NO FAILURE}}$

FIGURE 2–9 VIBRATION TEST COMPONENT 15 PCF (0.24 GM/CM^3) DYNAQUARTZ

2.3 FABRICATION

Fabricability of the design was proven by the successful fabrication of three RPP segments, two of which were incorporated in the deliverable assemblies pictured in Figures 2-1 and 2-2. Similarly, the insulation assemblies were proven fabricable, although design for employment of looser tolerances for the tiled configuration would markedly ease detail part fabrication and assembly. Experience gained on the leading edge assembly indicates that with suitable tooling and relaxed tolerances, the heatshield concept is fabricable for production quantities.

The RPP leading edge component varied in thickness from 34 plies (0.48 in. (1.22 cm) in the stagnation region to 16 plies at the upper trailing edge and 20 plies at the lower trailing edge. Rib flanges remained at 12 plies like the Phase II leading edges. To fabricate this design, debulking was required after laying up of every three to four plies and special debulking tools were required to obtain good compaction of skin/rib and rib/spar corner regions.

Processing of thick laminate components was adjusted from the previous thin laminate approach to eliminate delamination tendencies. The most significant factor of improvement was the introduction of extended post cure profiles to permit slower generation of decomposition gases. No delaminations in these thicker regions were experienced on any of the leading edges.

Coating of the leading edges required some adjustments from the Phase II technique because of the thicker skin panel. The first unit coated exhibited an undesirable panel coating thickness variation (roughly 2 to 1, being thicker on the outer surface) outside to inside, which was corrected with the introduction of graphite heat conductors to obtain more uniform temperatures on the part. On the second part coating thickness variation ranged from 0.020 in. to 0.033 in. (0.051 to 0.085 cm) on the front side and from 0.015 in to 0.033 in. (0.038 to 0.084 cm) on the aft side. With additional adjustment, coating thickness on the third part was more uniform, varying between 0.020 in. and 0.030 in. (0.051 to 0.076 cm).

2.4 COATING THICKNESS MEASUREMENT

Because the coating erodes during the course of 100 missions, it is desirable to assess coating thickness remaining by NDE techniques rather than by destructive evaluation of selected RPP segments. Three NDE approaches were examined: eddy current, beta backscatter, and ultrasonic-pulse-echo. VSD (eddy current and ultrasonic), Los Alamos Scientific Labs (beta backscatter), and Oak Ridge National Labs (eddy current) participated in the study. Beta backscatter and ultrasonic approaches proved unacceptable for achieving measurement accuracy for the material and thicknesses involved in the study.

Eddy current techniques in the 1.5 MHz range produced accuracies of ± 0.005 in. (0.013 cm) for a 95% probability of non-exceedence, while 5 MHz frequency equipment improved accuracy to ± 0.004 in. (0.010 cm). In view of the design criteria employed, which requires a theoretical 0.010 in. (0.025 cm) thick coating remaining at the life limit of the system, the measurement accuracy obtained by eddy current techniques is acceptable. Higher frequencies to 15 MHz should improve accuracy further.

An important feature of the Eddy current equipment is that it is lightweight and portable, which makes it equally convenient for shop or flight line use.

2.5 COATING CRAZING

Crazing of the coating, produced on RPP, has been inherent with the substrate/coating process employed. While all strength, physical property, and plasma data in the Phase II program were obtained with crazed material, and all data, including 100 mission thermal/oxidation cycling data, showed the material system to be acceptable for the 100-mission system life it was nevertheless desired to decrease crazing. The main reason for this was to increase resistance to subsurface oxidation, although the ability to obtain more definitive inspection acceptance criteria would be a by-product.

In comparative tests with standard processed materials, it was found that crazing could be reduced by the introduction of step heating to minimize thermal gradients, through additions of 0.5 - 1.0% silica to the coating pack, and more dramatically, by doping (0.5%) the pack with boron. Heating cycle variations and boron additions are worth further optimization and evaluation, because of their potential improvement to the system. Step heating was incorporated into the processing of deliverable hardware in this phase and becomes the new standard. VSD was supported in the coating crazing investigation by the University of Washington under a subcontractor effort.

2.6 DESTRUCTIVE EVALUATION PHASE II AND III LEADING EDGES

One leading edge from Phase II and one from Phase III were sectioned to examine strength and material appearance under magnification. These leading edges had undergone prior testing before sectioning. For example, the Phase II leading edge had been subjected to both boost pressure loading and entry heating, while the Phase III unit had been tested for 100 minutes in an acoustic environment of 163 db OA. The Phase II panel produced strengths equivalent to data obtained from flat panel testing. The Phase III leading edges generally exhibited equal or higher strengths than measured in flat panels. The significance is that material properties of components is at least equal to those values obtained from flat panel element testing.

2.7 MATERIAL PROPERTIES

Previous programs (References 1 through 3) determined material system performance on relatively thin, 13 ply material. As part of the Phase III effort, mechanical property data was gathered on coated RPP up to 34 plies (0.44 in.) (1.12 cm) in thickness.

Using 13 ply material strength as a basis for comparison, it was found that for the thicker laminates flexure, tension, shear and interlaminar properties were roughly comparable with the thinner material, but compression strength was lower for the thicker laminates.

Preliminary design properties for coated RPP derived from both Phases II and III test results are summarized in Table 2-1. These are shown to provide an indication of RPP performance, which of course is variable with temperature, thickness, and loading direction. Design strength values are nominally 67% of average test data to account for anticipated data scatter. This reduction factor was derived from a statistical analysis of Phase II flexure data to determine minimum ("A" values) strengths for design. All othér values shown are average test values.

TABLE 2-1 PRELIMINARY DESIGN VALUES COATED RPP

(For Reference Only)

Property (1)	13 Ply (1) (2)		34 Ply (1) (2)	
Strength, psi (N/m^2)				
Tension, warp	5700	(3930×10^4)	7000	(4830×10^4)
, fill	3000	(2070×10^4)	3800	(2620×10^4)
Compression, warp	7700	(5310×10^4)	4900	(3380×10^4)
, fill	7900	(5440×10^4)	4400	(3040×10^4)
Flexure, warp	9600	(6620×10^4)	11600	(8000×10^4)
, fill	8100	(5580×10^4)	8600	(5930×10^4)
Shear	2500	(1720×10^4)	2700	(1860×10^4)
Interlaminar Tension	300	(207×10^4)	400	(276×10^4)
Interlaminar Shear	1500	(1040×10^4)	1400	(965×10^4)
Elastic Modulus, psi (N/m^2)				
Tension, warp	1.9×10^6	(1.31×10^{10})	2.4×10^6	(1.66×10^{10})
, fill	1.5×10^6	(1.03×10^{10})	1.8×10^6	(1.24×10^{10})
Compression, warp	2.1×10^6	(1.45×10^{10})	2.7×10^6	(1.86×10^{10})
, fill	1.6×10^6	(1.10×10^{10})	1.8×10^6	(1.24×10^{10})
Shear	0.7×10^6	(0.48×10^{10})	0.8×10^6	(0.55×10^{10})
Density, lb/ft^3 (g/cc)	97	(1.56)	90	(1.44)
Emittance at 2500°F (1371°C)		0.94		
Specific Heat at 2500°F (1371°C) BTU/lb/°F (Joules/kg/°C) @ 2500°F (1371°C)		0.42 (176×10^{-3})		
Conductivity Parallel to Laminate, BTU-in/hr-ft^2-°F (Joules/hr-m-°C) @ 2500°F (1371°C)		100 (52000)		
Coefficient of Thermal Expansion in/in/°F (cm/cm/°C)		1.6×10^{-6} (2.9×10^{-6})		

(1) Values in parentheses are metric units

(2) Room temperature values shown unless otherwise noted.

3.0 CONCLUSIONS

As a result of the Phase III technology development program, the following conclusions were considered to be of major significance.

(1) Experiments show that thick laminate components are feasible for fail-safe designs and, although sensitive to processing, are insensitive to starting resin variations. Thick laminates do require tighter process control than thinner ones.

(2) Blockage of heat from the hot RPP (2500-3000°F) (1371-1649°C) to the cool aluminum (350°F) (177°C) is feasible through proper design of insulation and support brackets.

(3) Gap heating at seal strips is a result of suppressed cross radiation and surface roughness, producing surface temperatures that are manageable. Internal (cavity) heating is minimal, and poses no design/operational problems.

(4) RPP components generally exhibit material properties equal to or better than simple flat panels as determined by destructive evaluation.

(5) Because of consistency of shrinkage and expansion during processing, components can be fabricated within a ± 0.03 in. (± 0.076 cm) tolerance.

(6) Coating crazing, which is a normal consequence of VSD materials/processing, can be reduced by minor process changes and can be virtually eliminated by boron doping of the coating powders.

(7) Uniformity of coating thickness can be achieved by controlling heat distribution to the part in the coating pack.

(8) X-ray radiography and ultrasonic-pulse-echo provide effective NDE tools for in-process determinations and the internal quality of RPP components. These techniques are applicable at various stages of processing from initial cure through coating.

(9) Coating thickness can be measured by NDE techniques to an acceptable tolerance level ($\pm$0.004 in.) ($\pm$0.010 cm) through the use of eddy current equipment, which is light-weight and portable for shop and flight line use.

(10) RPP and associated insulation components can be designed to meet the severe acoustic noise and vibration dynamic load environments experienced by the Shuttle.

4.0 RECOMMENDATIONS

The Phase III program resulted in identification of additional studies to advance the RPP technology for either improved performance, reduced cost, better inspection or increased understanding of the material system/design. The following studies are recommended to further RPP technology.

(1) Investigate the use of higher frequency (5-15 MHz) eddy current equipment to improve coating thickness measurement accuracy below the current $\pm$0.004 - 0.005 in. ($\pm$0.010 - 0.013 cm) capability.

(2) Pursue coating modification with boron doping to reduce coating crazing further, thereby enhancing subsurface oxidation resistance.

(3) Examine reduced resin content RPP material further as a means of improving fabricability and gaining better tolerance control. This would enhance debulking during layup, and would reduce resin char and associated shrink/growth.

(4) Perform more extensive mechanical property tests, optimizing specimen configuration and test technique to reduce data scatter. This effort should include round robin testing involving several test agencies. As an adjunct to this, RPP processing controls should be examined further to ensure satisfactory reproducibility.

(5) Conduct additional gap heating tests with more extensive thermocouple instrumentation and possibly with infrared photography to gain better thermal mapping in the seal strip region. These tests should include coated RPP for more exact temperature distribution, and higher pressures to assess lower altitude performance.

(6) Develop methods of assuring tight control of tolerance during manufacturing.

REFERENCES

(1) Development of a Thermal Protection System for the Wing of a Space Shuttle Vehicle, Phase I Final Report, LTV/VSD Report No. TR143-5R-00044, no date (NASA Contract NAS9-11224).

(2) Development of a Thermal Protection System for the Wing of a Space Shuttle Vehicle, Phase II Final Report, LTV/VSD Report No. T143-5R-00124, March 1972 (NASA Contract NAS9-11224).

(3) Oxidation Inhibited RPP Space Shuttle Belly Panel, LTV Report No. T159-5R-00047, dated 31 January 1972.